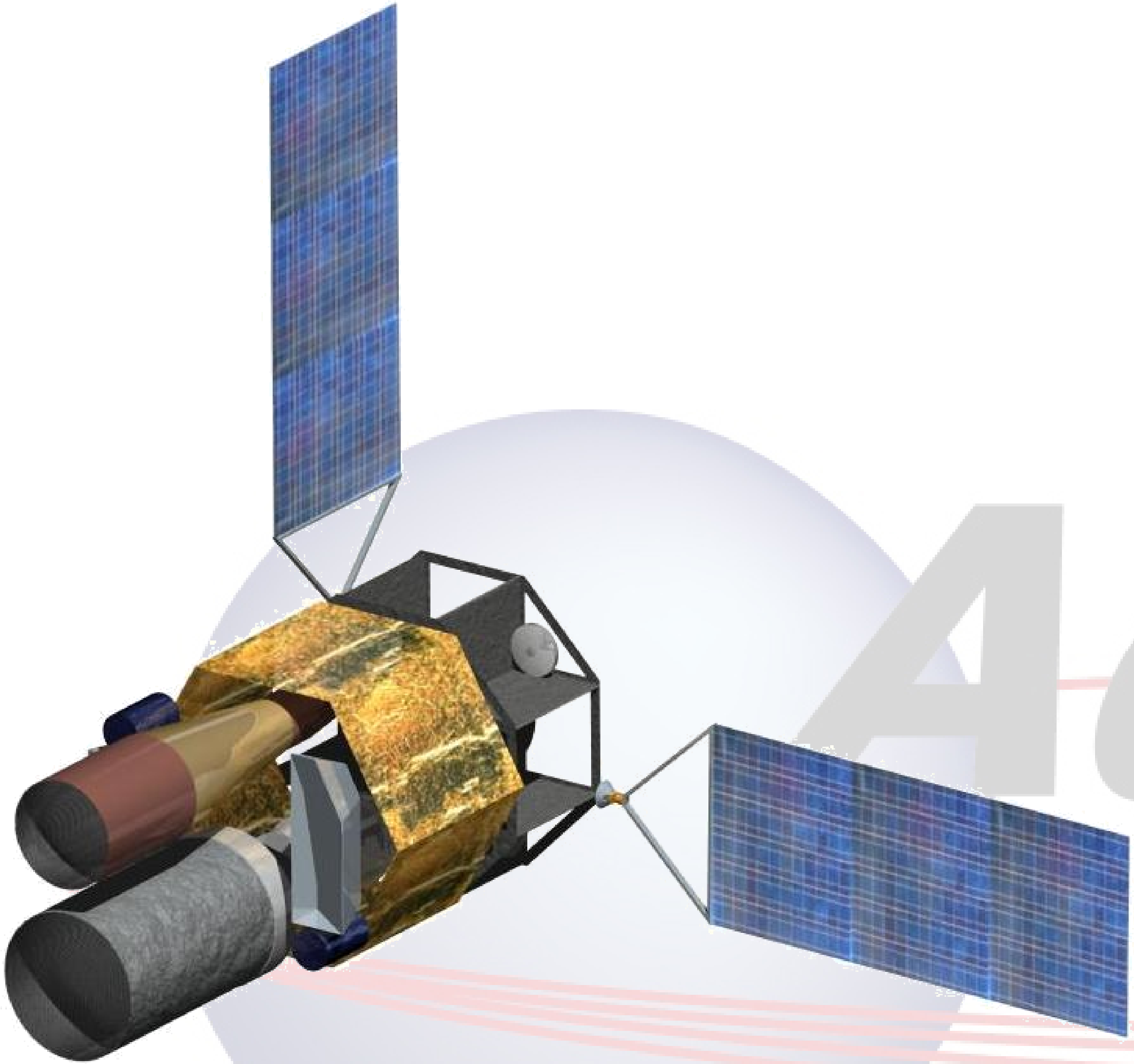
Xenia Spacecraft Study



Randy Hopkins

October 26, 2008

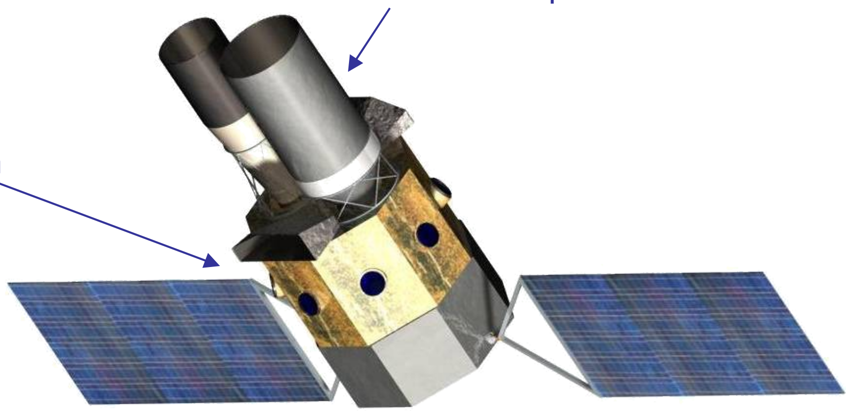
Xenia Spacecraft Study Approach

Randy will provide study approach chart here

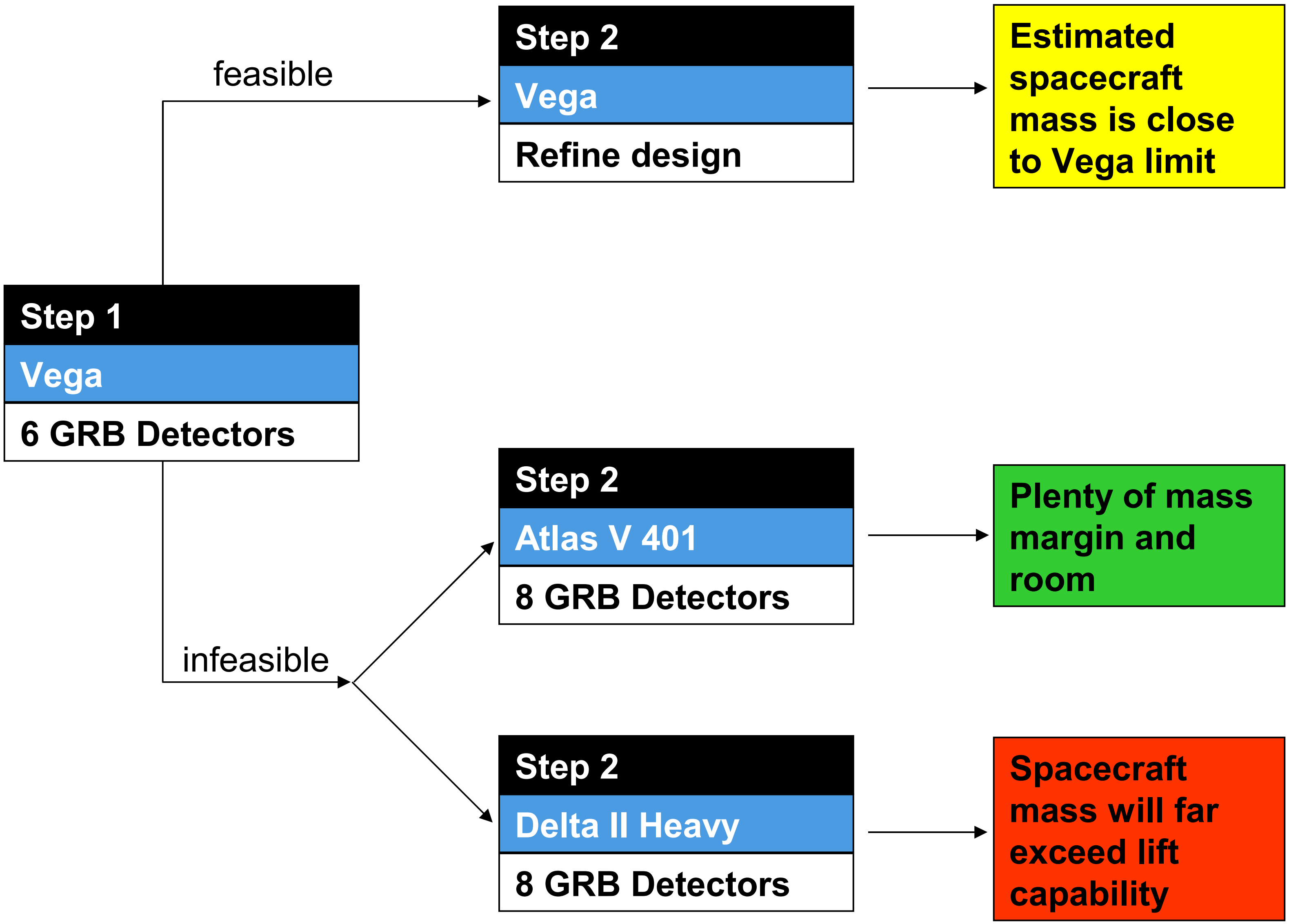
Study Overview

- **Goal**
  - Perform a mission concept study for the proposed Xenia mission

- **Responsibilities**
  - Spacecraft: ED04
    - Avionics / GN&C
    - Communications
    - Electrical Power
    - Trajectory / Mission Analysis
    - Propulsion
    - Science Instruments Integration
    - Launch Stack Shroud Integration
  - Science: VP62
    - Science Instruments Definition
    - Science Instruments Design
    - Mission requirements



Roadmap and Status



Mission Analysis

- **Target orbit: 600km circular, inclination equal to 5 degrees**
  - Min Dark Period 35.5 min
  - Max Light Period 61.2 min
- **Orbit lifetime: 10 yrs**
- **Spacecraft lifetime: 5 yrs**
- **De-orbit**
  - Re-entry interface defined as 400000ft altitude (122 km)
  - Circular orbit altitude: 600 km
  - Target re-entry flight path angle: -1.75 degrees
    - Acceptable range: -1.4 to -2 degrees
      - Reference Hubble De-orbit Study
  - Thrust tangent to velocity vector
  - Impulsive and Finite Burn maneuvers modeled with Copernicus

Xenia Spacecraft Study
Ground Rules & Assumptions

Attitude and Orbit Control

- Autonomous slewing to a transient event (detected by WFM) with a goal slew rate of 1 deg/sec
- Acquisition of new transient event locations with an accuracy of 2' within a time frame of 60-120 sec for a source 60° off the original pointing
- Pointing: 3 axis stabilized
  - Better than 2', knowledge < 1"
- Pointing accuracy (absolute pointing drift) on longer time scales (>5 min) in the range of 0.75' to 1.25'
- Post-facto pointing resolution <3"

Propulsion

- **Deorbit DV 163 m/sec (from mission analysis)**
- **T/W > 0.025**
- **Liquid propulsion**
  - 2 spherical tanks for each propellant (bi propellant)
  - Pressure-fed system
    - Separate pressurant system for bipropellants (vapor creep issue)
  - 1 fault tolerant
- **Solid propulsion**
  - Off-the-shelf solid rocket motor from ATK catalog
  - Minimize off-loading

Avionics

- Pointing/viewing coverage: 360deg (entire sky), with 45deg sun avoidance, earth, moon avoidance
- 1st fast slew on GRB: 60deg /60sec (i.e.1deg/s), once per 24 hr (estimated)
- Pointing accuracy: 2’ within 60-120 sec, knowledge < 1”
- 2nd pointing resolution: <3” (EDGE), within 20s (estimated)
- Slow slew on command: Rate = 0-4 deg/min, once per 24 hr period
- Drift: 0.75’ to 1.25’ over 5 minutes (EDGE)
- Telemetry: 3.8 Mbps

Xenia Spacecraft Study
Ground Rules & Assumptions

Power

- **Solar Arrays**
  - GaAs 4j 348 W/m^2 before knockdowns, 3% / year degradation
    - Must be pointed independently of detectors
  - 2.24 kg / m^2
  - Inherent Degradation 0.85
  - Degradation Rate 0.03/yr
- **Secondary Batteries**
  - Li-Ion, high number of cycles, 40% DOD
- **Conditioned power**
  - multiple voltages from common power bus @ 28V

Thermal

- Assuming experiments have assessed their thermal control requirements. Power for needed heaters and coolers is considered in the experiment power estimates. Heat rejection of experiment power will be accommodated by spacecraft radiators.
- Need power breakdown from avionics to determine heater requirement during eclipse time and radiator requirement during sun time.
- Need spacecraft equipment temperature constraints (-30C to 40C) to start.
- Need basic geometry configuration of spacecraft and experiments to determine MLI mass, and orbital average boundary temperatures for heater sizing.

Xenia Spacecraft Study
Ground Rules & Assumptions

Thermal

- Primary objective is to develop a thermal design concept for the Xenia spacecraft
- Circular orbit, 5° inclination
- βmax = 33.5 °, βmin = 0 °
- 3-axis stabilized, Articulated solar arrays
- 45° sun avoidance angle.
- Assume the instruments have assessed their thermal control requirements and are thermally isolated from the spacecraft bus.
- Assume power for instrument required heaters and coolers is included in the instrument power estimates.
- Assume mass for instrument thermal control is included in the instrument mass estimate.
- Heat rejection of instrument power will be accommodated by spacecraft radiators.
- Spacecraft subsystems heater requirements will be sized for worse cold case, radiator area will be sized for worse hot case.
- Total max power estimate = 1666W

	WFS	WFI	WFM	GRB M	Spacecraft
Mass (kg)	389	245	133	300	TBD
Power (W)	620	140	175	152	~400
Max power (W)	679 (1)	160 (2)	175	152	~500

(1) Heat dissipation (430W) for cryogenic-free cooler for CCD detector may require dedicated radiator, heatpipes, thermal doubler, and/or phase change device. Mirror temperature is controlled using 125W heater power and a long thermal baffle. Electrical heater network dissipates heater power directly on the optics.

(2) Detector passively cooled -50C and further cooled to operating temperature of -80C using Thermal Electric Cooler. Operational temperature of the mirror system is 20°±5 C achieved by use of a warmed baffle (100 cm) with thermal filter. Heater power of 40W.

Structures

- **Maximum Launch Loads**
  - Vega: payloads > 300 kg to 2100 kg (adapter 60 kg)
    - 5.5 g along launch axis
    - .9 g lateral to launch axis
  - Atlas V 401: payloads to 4390 kg (adapter & req HW 120 kg)
    - 5 g along launch axis
    - 2.0 g lateral to launch axis
  - Delta II Heavy: payloads to 900 kg

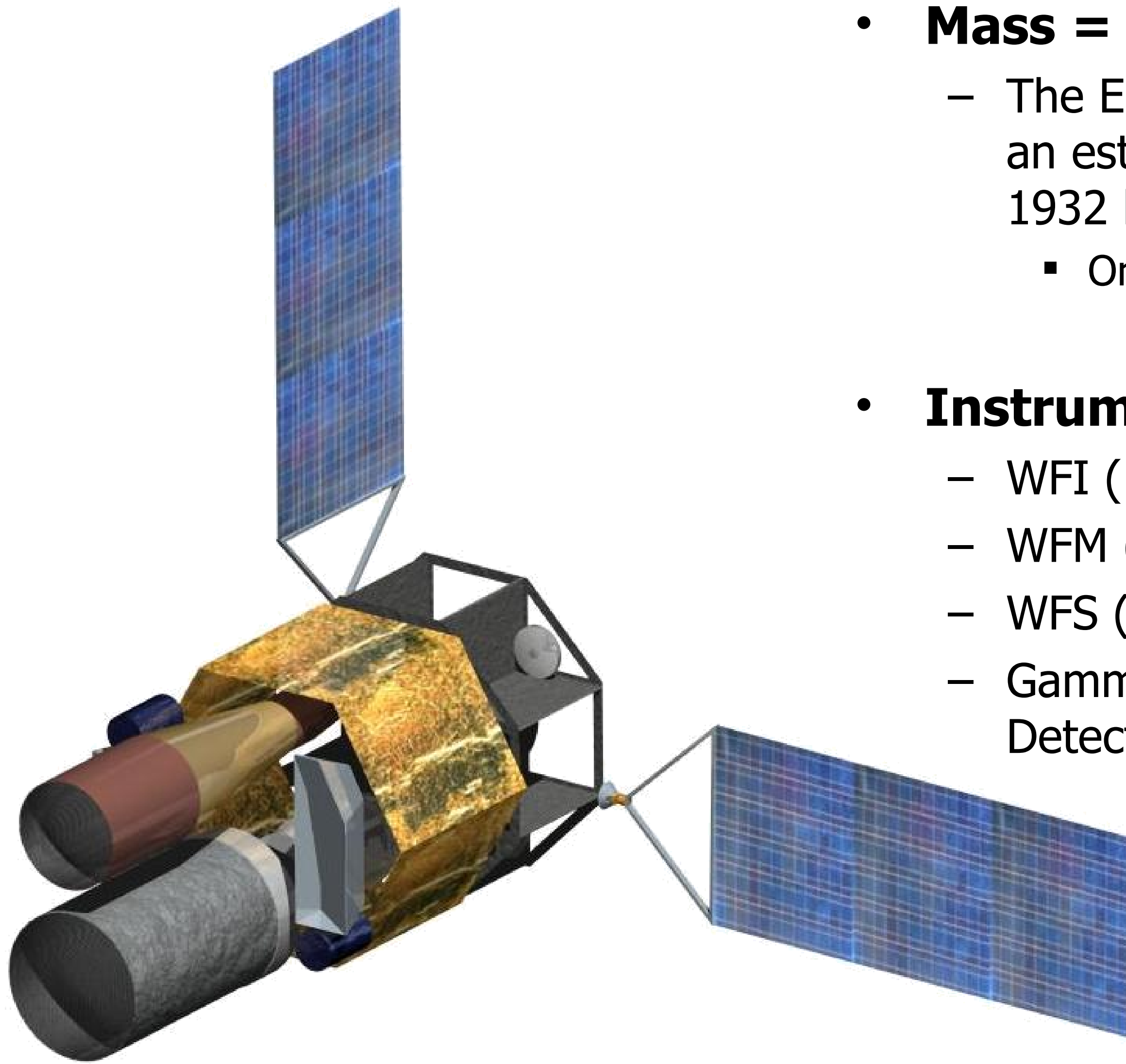
Results and Initial Design

Launch Vehicle Performance Summary

	Vega	Atlas V 401	Delta II Heavy (7920H-10)*	Delta II Heavy (7925H-10)* [3]
Launch Site	Kourou	KSC	KSC	KSC
Source:	Vega User's Manual [1]	NASA LSP [2]	NASA LSP [2]	NASA LSP [2]
600 km @ 5 deg	2110	4395	895	NR
600 km @ 10 deg	2100	5815	1440	NR
500 km @ 5 deg	2180	4390	885	NR
500 km @ 10 deg	2170	5820	1435	NR

- NOTES
  - * Also known as the 2920H-10 and 2925H-10.
  - [1] Interpolated results from performance plots.
  - [2] The Atlas and Delta II guides do not include performance estimates for these low inclination missions. These data are direct quotes from NASA Launch Services Program.
  - [3] Includes a Star-48 third stage, which adds about 30% to the payload capability for these inclinations.

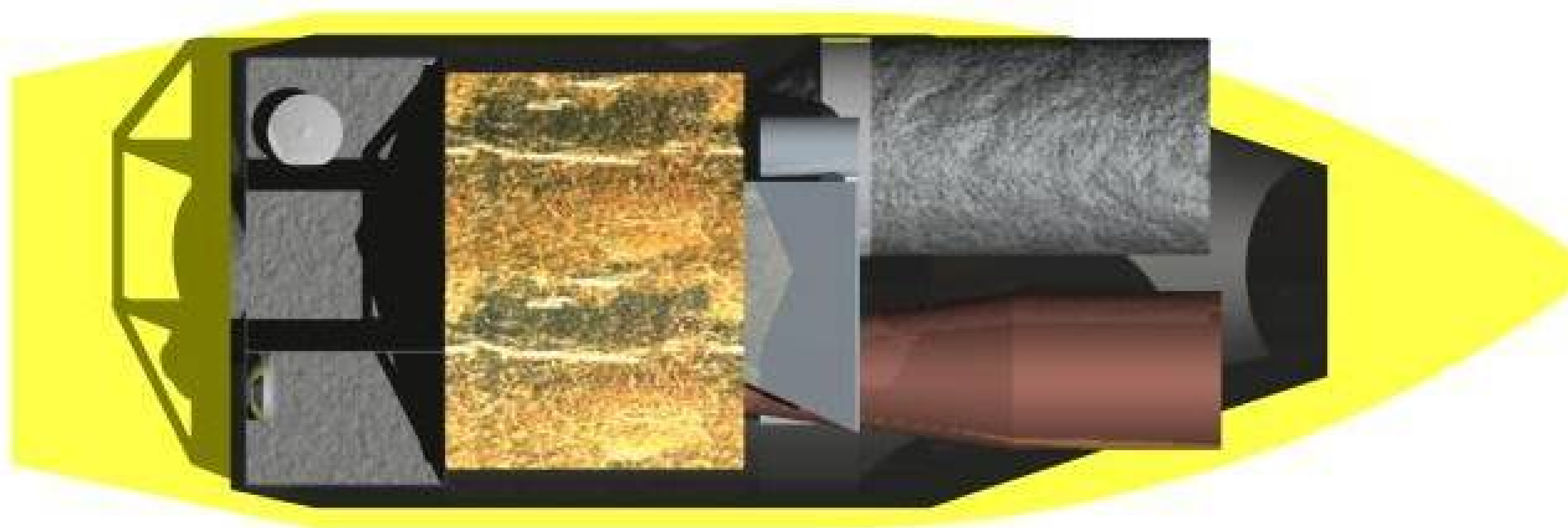
Current Spacecraft Design / 6 GRB Detectors / Vega



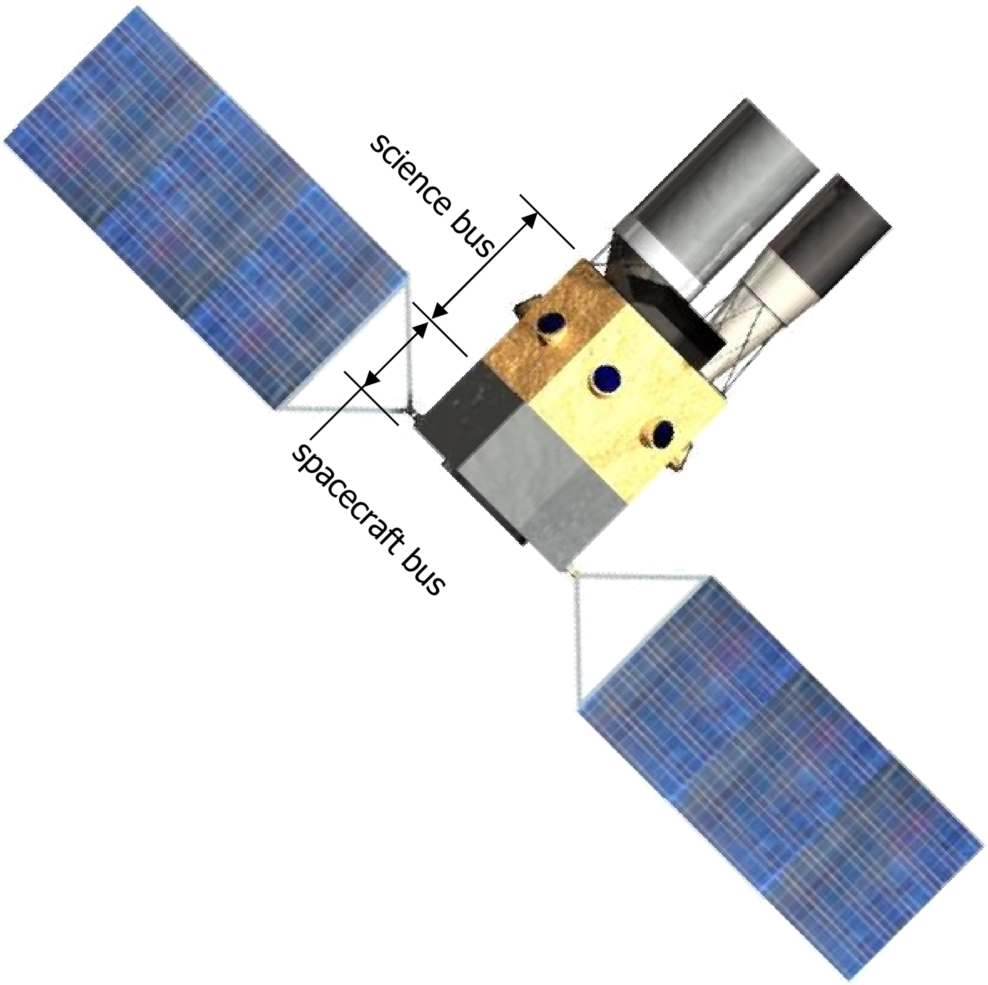
- **Mass = TBD**
  - The EDGE proposal listed an estimated mass of 1932 kg
    - Only includes 2 GRBD
- **Instruments**
  - WFI (1)
  - WFM (1)
  - WFS (1)
  - Gamma Ray Burst Detector (6)

Current Spacecraft Design / 6 GRB Detectors / Vega

- Using a rough estimate of the volume for various subsystems, the spacecraft does not appear to fit within the volume constraints of the Vega shroud
- Design must be completed before we can make a final determination

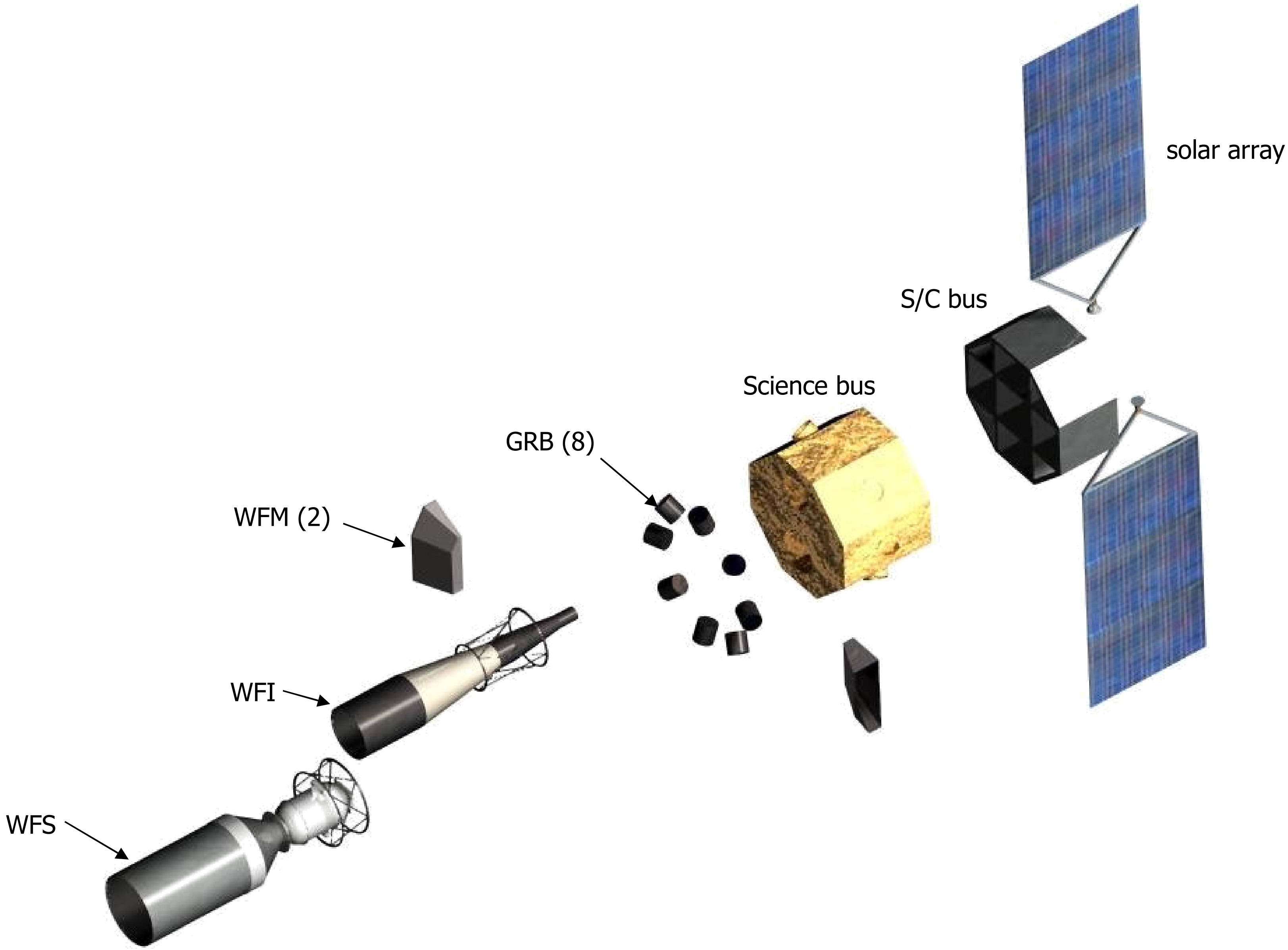


Current Spacecraft Design / 8 GRB Detectors / Atlas



- **Mass = TBD**
  - The EDGE proposal listed an estimated mass of 1932 kg
    - Only includes 2 GRBD
- **Instruments**
  - WFI (1)
  - WFM (1)
  - WFS (1)
  - Gamma Ray Burst Detector (8)

Current Spacecraft Design / 8 GRB Detectors / Atlas



Current Spacecraft Design / 6 GRB Detectors / Atlas

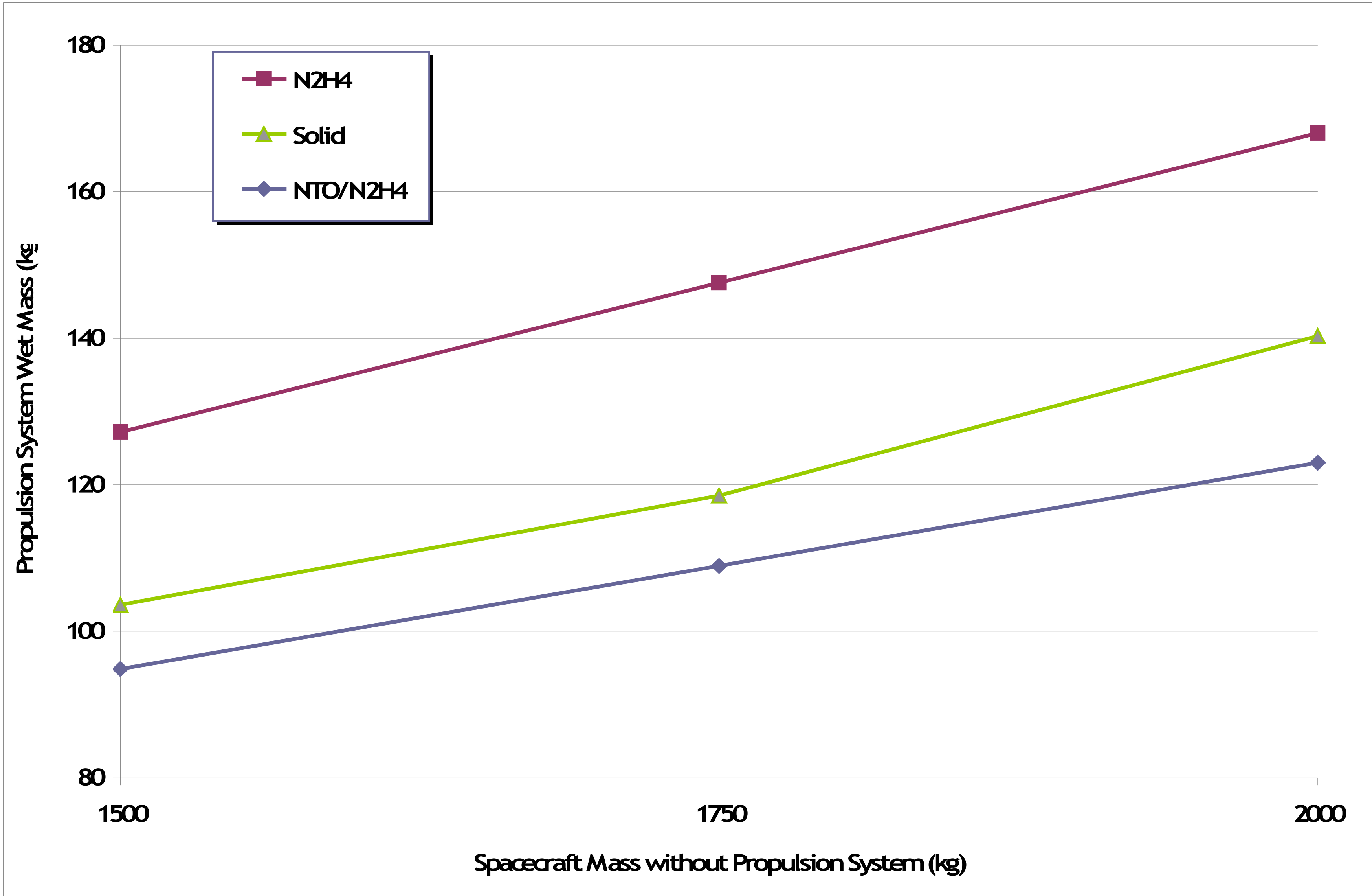
- Using a rough estimate of the volume for various subsystems, the spacecraft easily fits within the volume constraints of the Atlas V 401 shroud
- Plenty of mass margin as well

Xenia Spacecraft Study
Initial Results: Deorbit

- **Method**
  - Impulsive and Finite Burn maneuvers modeled with Copernicus
- **Results**
  - **Delta-V = 163 m/s** for a reentry flight path angle of -1.75 degrees
    - Impulsive Delta-V: 161.3 m/s
    - Gravity Loss: 1.7 m/s (assuming worst case T/W = 0.025)
    - Margin: 0 m/s (assumptions are already conservative)
  - Perigee altitude = 34.6 km
    - Ranges from 65.7 km to 8.25 km for the acceptable range of reentry flight path angles
  - **Gravity Loss** is insignificant for T/W > 0.025

Xenia Spacecraft Study
Initial Results

Propulsion



Xenia Spacecraft Study
Initial Results

Power Masses		Qty		215.50 kg
	PDU	1	11.59 kg	11.59 kg
5 m, redundant	Cabling	1	8.79 kg	8.79 kg
	ARU	1	41.95 kg	41.95 kg
	Solar Array	1	34.05 kg	34.05 kg
5184 Wh	Secondary Battery	3	11.66 kg	34.98 kg
	Battery Charger	1	84.14 kg	84.14 kg

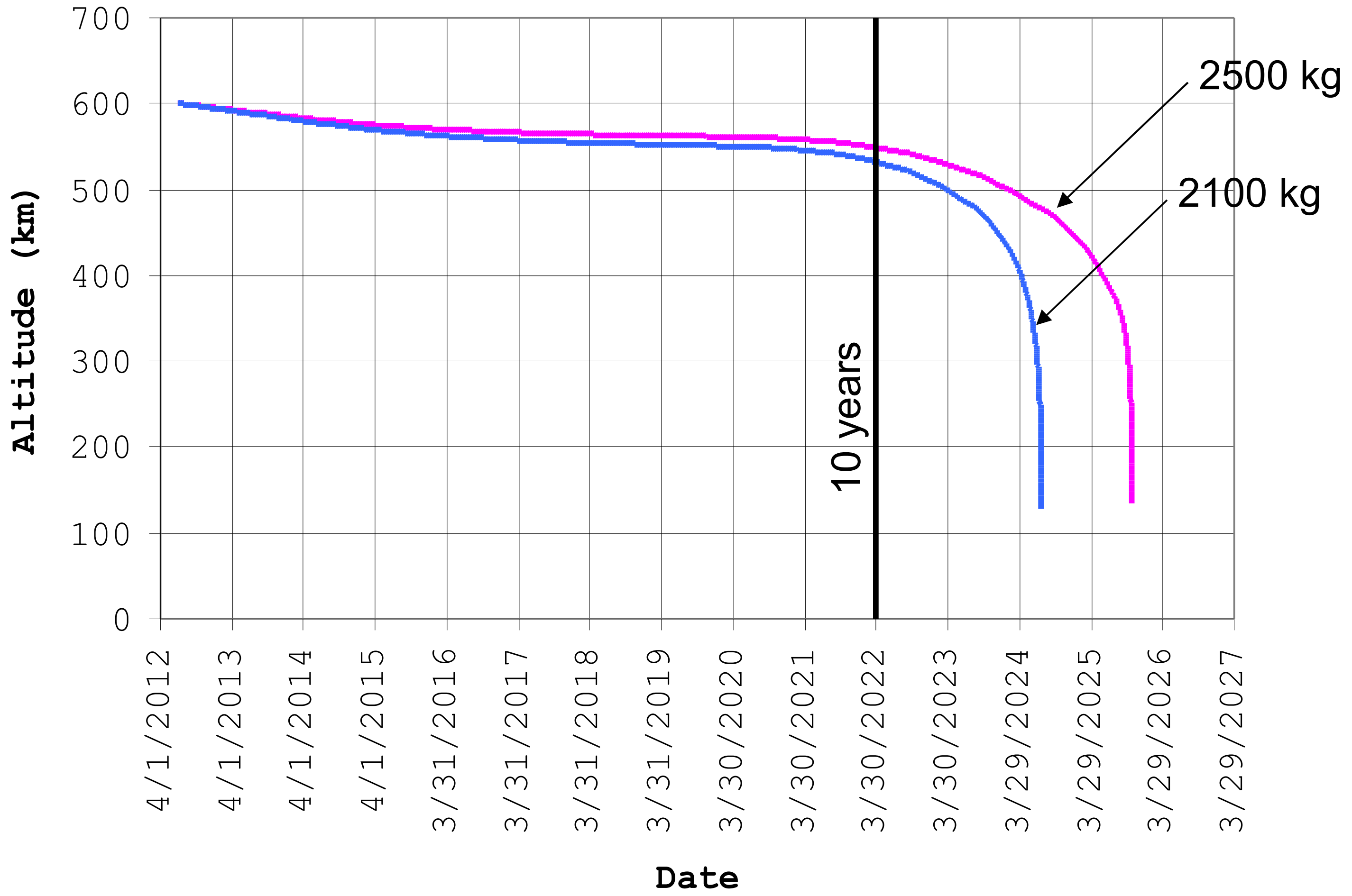
- Solar Array – 15.20 m^2
- Secondary Batteries – 8 Cells per Unit, 3 Units
  - Based on Saft Li-Ion VES 180 Cells (50 Ah de-rated to 45Ah, 3.6V)
- Array Regulation – Direct Energy Transfer (0.95 Efficiency)

Xenia Spacecraft Study
Initial Results

Structures

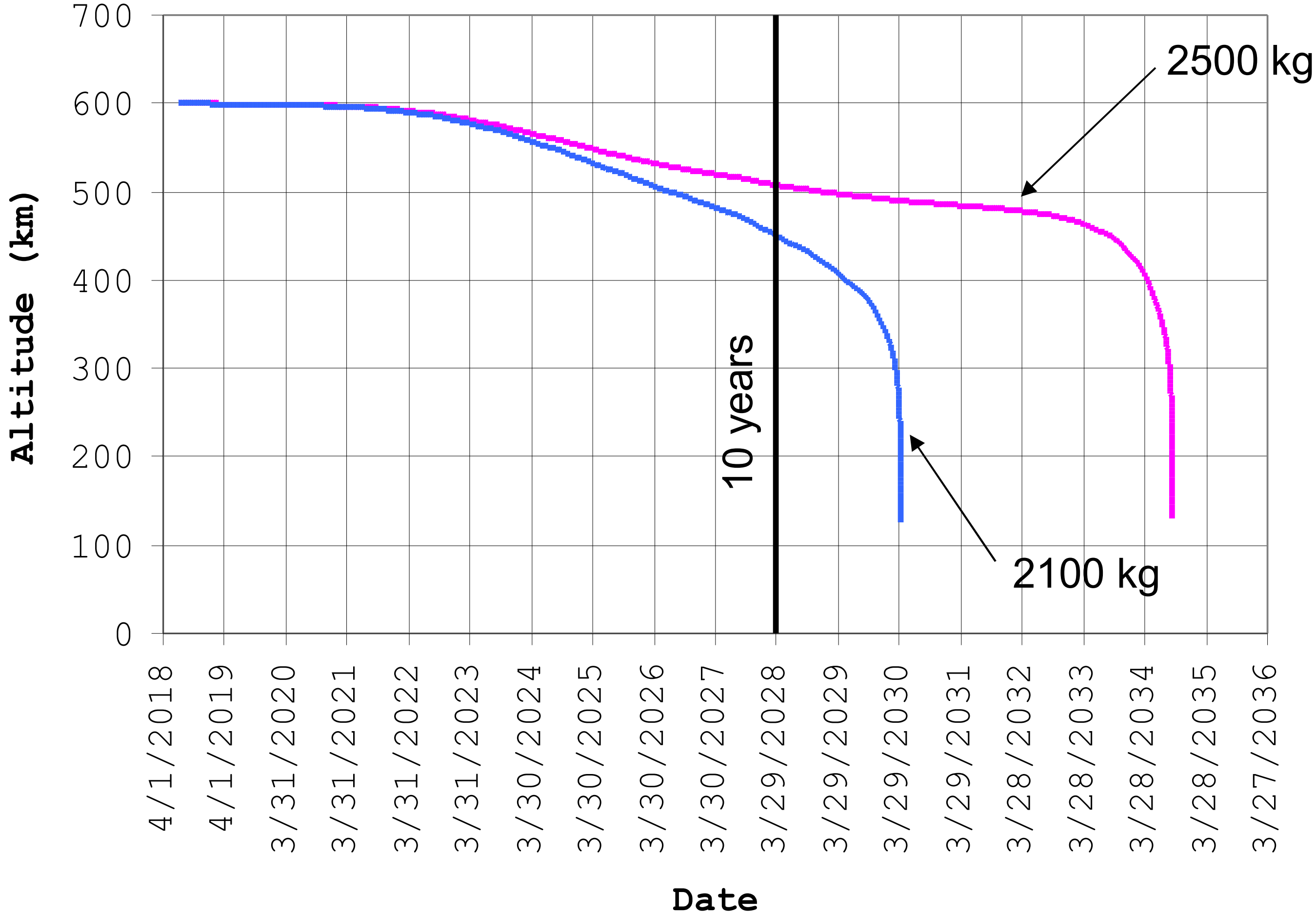
- **Materials of construction**:
  - Truss or vehicle body: Aluminum 7075, 2219, 6061 are candidates
  - A-286 stainless steel fastening hardware, some structural members
- **Solar panels**:
  - Estimated weight for two panels: 36 kg for 15 sq meters
  - 20 kg for boom and secondary structure, including folding mechanism
  - 17 kg for motor and articulation gears to enable panels to track sun

Orbital lifetime: 2012 Launch



Xenia Spacecraft Study
Initial Results

Orbital lifetime: 2018 Launch



Orbital lifetime: conclusion

- **Based on the preliminary analysis done using STK:**
  - No propulsion system is needed to periodically boost the satellite's orbit and achieve the target 10-year orbital lifetime

- **Reevaluate once the initial spacecraft design is completed**

Backup

Xenia Spacecraft Study
Ground Rules & Assumptions

- **Performance: Vega**
  - Sufficient propellant reserve in AVUM to reach the targeted orbit with a 99.7% probability except otherwise specified. The AVUM's fuel capacity is also sufficient for deorbitation or for transfer to a safe orbit as required.
  - Aerothermal flux at fairing jettisoning and second aerothermal flux is less or equal to 1135 W/m^2. Increasing this value would improve LV performance by allowing an earlier fairing jettisoning or adapting the ascent profile.
  - Altitude values are given with respect to a spherical earth radius of 6378 km.
  - The orbital flight realized with standard attitude sequence and duration, with standard telemetry provisions and electrical services to the spacecraft,
  - The flight path takes into account the relevant CSG safety requirements.

  *Source: *Vega User's Manual*

- **Performance: Delta II Heavy**
  - This performance does not include the effects of orbital debris compliance, which must be evaluated on a mission-specific basis. This could result in a significant performance impact for missions in which launch vehicle hardware remains in Earth orbit.
  - 99.7% Probability of Command Shutdown (PCS)
  - 6915 Payload Attach Fitting (PAF)
  - Park orbit perigee = 185 km (100 nmi)

Xenia Spacecraft Study
Ground Rules & Assumptions

- **Performance: Atlas V**
  - **This performance does not include the effects of orbital debris compliance, which must be evaluated on a mission specific basis. This could result in a significant performance impact for missions in which launch vehicle hardware remains in Earth orbit**
  - **3-sigma mission required margin, plus additional reserves determined by the LSP.**
  - **Launch from SLC-41 at CCAFS (Cape Canaveral Air Force Station).**
  - **For LSP, the Type B2 payload adapter was originally assumed. This adapter is currently no longer used; however one of similar mass has become the baseline. The performance shown here is not affected by this change.**
  - **4-meter Extended Payload Fairing (EPF).**
  - **Max sustained acceleration is 5g's, occurring near the end of the first stage burn**
  - **Booster stage is throttled to enforce the 5g limit**
  - ***Separated spacecraft mass does not include payload adapter mass and other mission peculiar hardware which may be required***

Xenia Spacecraft Study
Ground Rules & Assumptions

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Orbital lifetime: inputs to model

Model Parameters		
Atmosphere	NRL MSISE 2000	
Solar flux sigma value	2	
Rotating Atmosphere	no	makes calculations more conservative
Satellite Parameters		
Drag coefficient	2.2	
Drag area	24	m^2
Area exposed to sun	30	m^2
reflection coefficient	1	
Satellite mass	2100 – 2500	kg

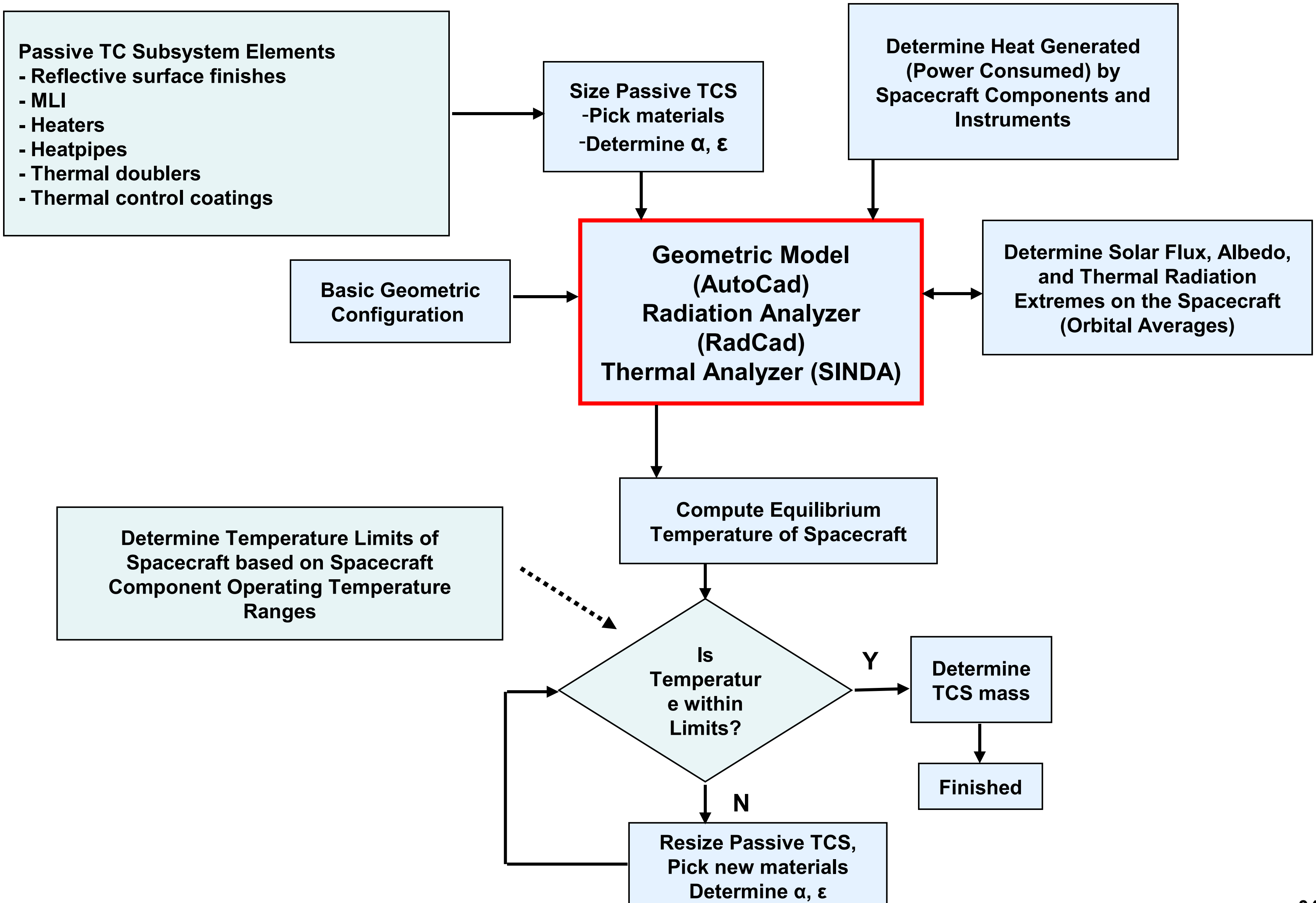
Propulsion Approach

- **Evaluate three alternative propulsion systems**
  - Liquid
    - Bipropellant (NTO/N2H4)
    - Monopropellant (N2H4)
  - Solid
- **Determine performance impacts of each propulsion option**
  - Propulsion system wet mass
  - Volume (tanks and SRM)
- **Understand issues with each option**

Propulsion Path Forward

- **Need to understand overall spacecraft mass**
- **Develop propulsion system configuration**
- **Provide inputs to other disciplines**
  - Thermal – size/mass of propellant and propellant tanks
  - Configuration – size of propellant and pressurant tanks
- **Develop issues to carry forward during spacecraft design**

XENIA Thermal Subsystem Design Concept



XENIA
Thermal Subsystem Design Concept

- Design for coldest permissible range (cold bias payloads over 1000 kg) and use heaters to trim.
- Thermal control surfaces → low absorptivity, high emissivity.
- Insulate as needed to stabilize temperature oscillations and ideally utilize equipment power as heat.
- Radiator panels located as needed on the sides (anti-sun) and bottom of the subsystems and instrument bus.
- Diode heat pipes may be required as thermal switches for disconnection in case radiators are exposed to the sun or earth.
- Use internal spacecraft bus environment (-20°C /50°C) until better numbers are available.

Thermal Design Temperature Limit for Typical Spacecraft Component Operating Ranges (° C) Min/Max			
Communications	Receiver/Multiplex	+10/+30	
Power System			
Attitude Control			

To be completed

CMG vs Reaction Wheels

CMG Advantages and Disadvantages

Advantages:

- CMGs provide torque amplification, making them much more efficient over reaction wheels.
- They require much less power than reaction wheels because of this efficiency, less than 10%.
  (however, can be equivalent in mass due to complexity, see EDGE charts 17,18)
- They can provide high torque values for fast slewing maneuvers.

Disadvantages:

- All CMG configurations have singularity states, in which no torque can be provided.
  (when all wheel torque vectors are perpendicular to the commanded direction, ref 1, pg 440)
- Singularity avoidance algorithms are complex and immature.

CMG TRL maturity

- The EDGE program decided to not use CMG because of immaturity of the control theory and because no CMGs had been flown on a commercial satellite at that time. (EDGE proposal, Piro, pg 28, and ref 2 pg 1)
- Since then, Ball Aerospace has flown a CMG on their Worldview satellite (a mapping satellite launched 9/07), and plans to launching another in 2009. (Control moment gyroscope, Wikipedia)
- This being the case, It’s suggested we give CMGs a try on Xenia.

CMG vs Reaction Wheels

CMG vs. Reaction Wheels

A CMG is a torque amplifier. A small torque on the gimbal produces a large torque on the spacecraft [1]:

$$\mathbf{T_{CMG} = I_w\omega} \times \boldsymbol{\delta_g}\mathbf{/dt} \quad ^{(2)}$$

where the wheel momentum $I_w\omega$ is amplified by the gimbal velocity of δ/dt

ω = wheel angular velocity

δ = gimbal angle

while reaction wheels simply produce torque per:

$$Tw = I_w\omega/dt \quad ^{(2)}$$

which also = $I_s\theta/d^2t$

I_s = spacecraft moment of inertia

θ/d^2t = spacecrafts angular acceleration

In summary:

In reaction wheels, torque has to be generated by accelerating the wheels. If you want lots of torque, you need lots of acceleration of massive wheels, which requires lots of motor power.

With CMGs, the momentum energy is already there in the wheels because of the constant high speeds, you just need to redirect the momentum to get torque, which can be done with relatively little gimbal motor power.

References:

1) *Space Vehicle Dynamics and Control*, AAIA, Bong Wei

2) *Attitude Control for Small satellites using Control Moment Gyros*, Lappas, Underwood

CMG vs Reaction Wheels

Possible CMG Configurations:

•SGCMG - Single Gimbal Control Moment Gyro. Used for fast, high torque, low power maneuvers. At least 4 wheels are needed to avoid singularities, usually in a pyramid configuration, Fig. 2-2(a). An additional wheel at the base (4 + 1) can be used for additional singularity avoidance and redundancy. This configuration was proposed in the EDGE study.

•4-SGCMG pyramid systems have an advantage of providing a spherical momentum envelope, equal momentum capability in all three axes. (ref 2 pg 1)

•Some other SCCMG configurations are shown in Fig. 2-2. Symmetric type S(6) was used on the MIR space station. Pyramid type S(4) is the most studied type because of its singularity avoidance capability using only 4 wheels.

•DGCMG - Double Gimbal Control Moment Gyro. Used for momentum storage (attitude control), station keeping of large space stations. Used on ISS. They require more power because torque of one gimbal usually requires a reaction form the other gimbal. Chances of hitting singularities are greatly reduced because of the double gimbal action.

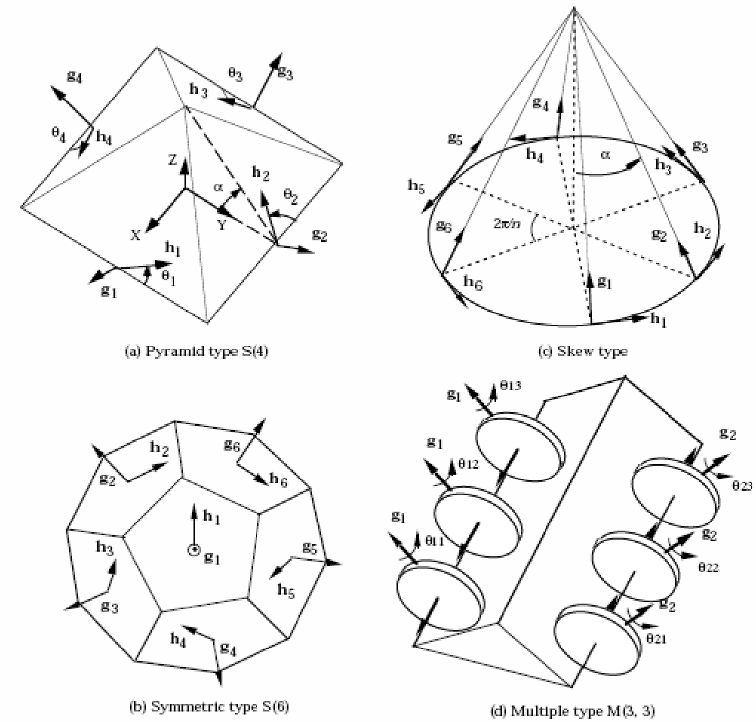


Fig. 2–2 Configurations of single gimbal CMGs

CMG vs Reaction Wheels

CMG Singularity Summary:

All CMG configurations have singularity states, in which no torque can be provided.

Example singularity O/P torques for a pyramid 4 wheel configuration:

elliptical	d = (-90, 0, 90, 0) deg
hyperbolic	d = (90, 180, -90, 0) deg

Using the most efficient CMG steering law (the pseudo-inverse steering logic), CMGs will tend to drift towards singularity states over time. (ref 1 pg441)

At least 4 wheels are required to avoid singularities using any configuration.

There are 3 kinds of established singularity avoidance logic:

1) Local avoidance logic - cannot guarantee singularity avoidance.

2) Global avoidance logic - will require intensive computations, which compromises real-time control.

3) Null motion - Null motion is gimbal motion that produces no net control torque on the spacecraft. It is accomplished by having the wheels react against each other. This action causes a creep towards saturation. Null motion can eliminate hyperbolic singularities, but not elliptical singularities.

Some new singularity avoidance algorithms:

1) Escape/Avoidance steering logic, Bong Wie, Honeywell International Inc, July 2005.
Based on a mixed weighed two-norm least squares optimization. Overcomes saturation singularities and avoids internal elliptical singularities.

2) Continuous CMG control, David Bailey et al, Honeywell International Inc, Oct. 2000.
Uses both closed and open loop control of CMG. Detects when pseudo-inverse rule will cause a singularity, then switches to an open loop control until the singularity is passed.

3) Kennel's steering law, used on the ISS with double gimbaled CMG system. Still possible to hit singularities, but chances are greatly reduced because of the double gimbal and no gimbal stops.

Comments:

Don't know which control law algorithm Ball Aerospace used for their Worldview satellites at this time.

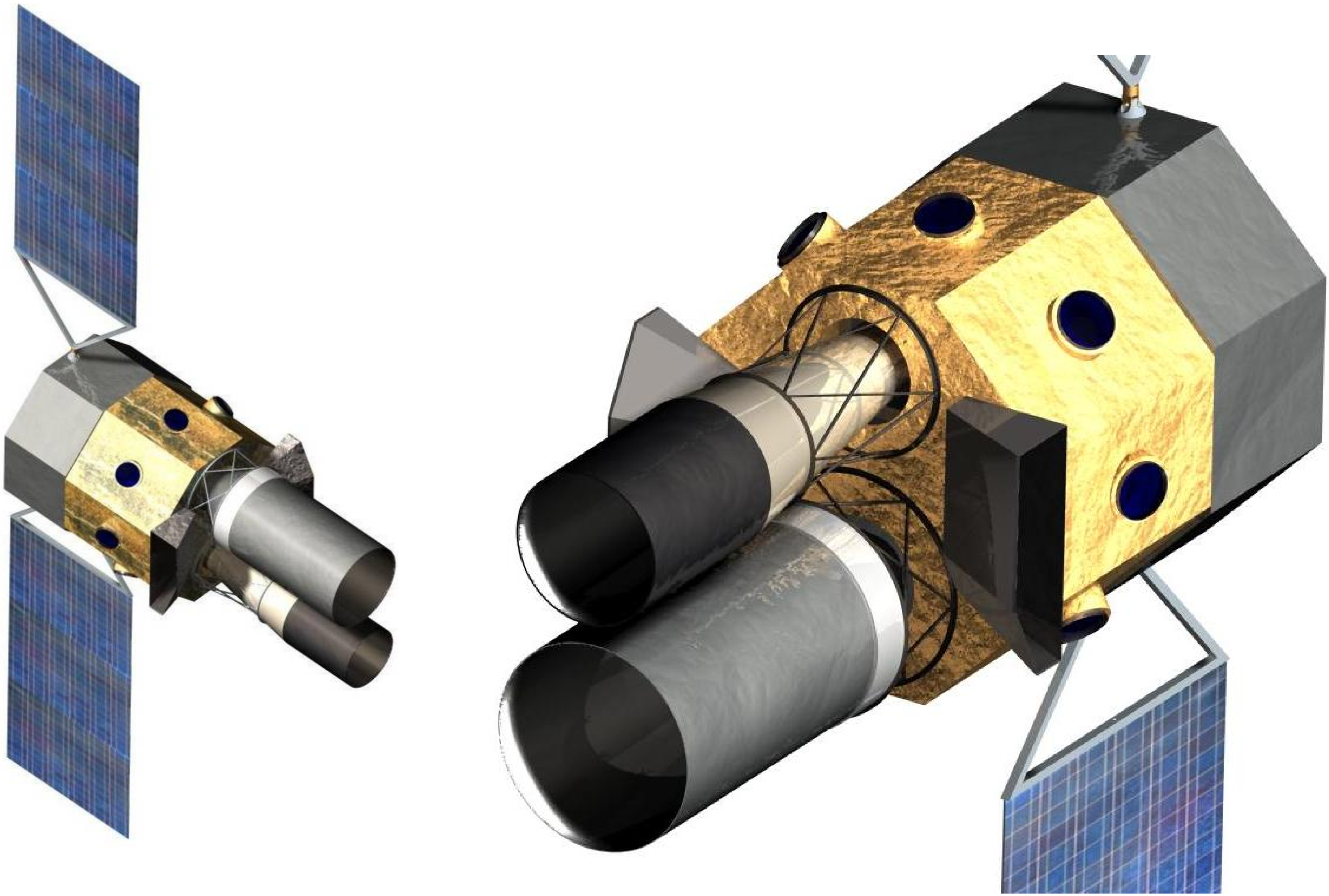
Power
Design Highlights

- **Solar Array – 15.20 m^2**
  - GaAs 3j rated 348 W/m^2 (before Knockdowns)
  - 2.24 kg / m^2
  - Inherent Degradation 0.85
  - Degradation Rate 0.03/yr

- **Secondary Batteries – 8 Cells per Unit, 3 Units**
  - Based on Saft Li-Ion VES 180 Cells (50 Ah de-rated to 45Ah, 3.6V)
  - 1.29 Packing Factor
  - Cell Load Balancing Electronics
  - Max Depth of Discharge < 40%

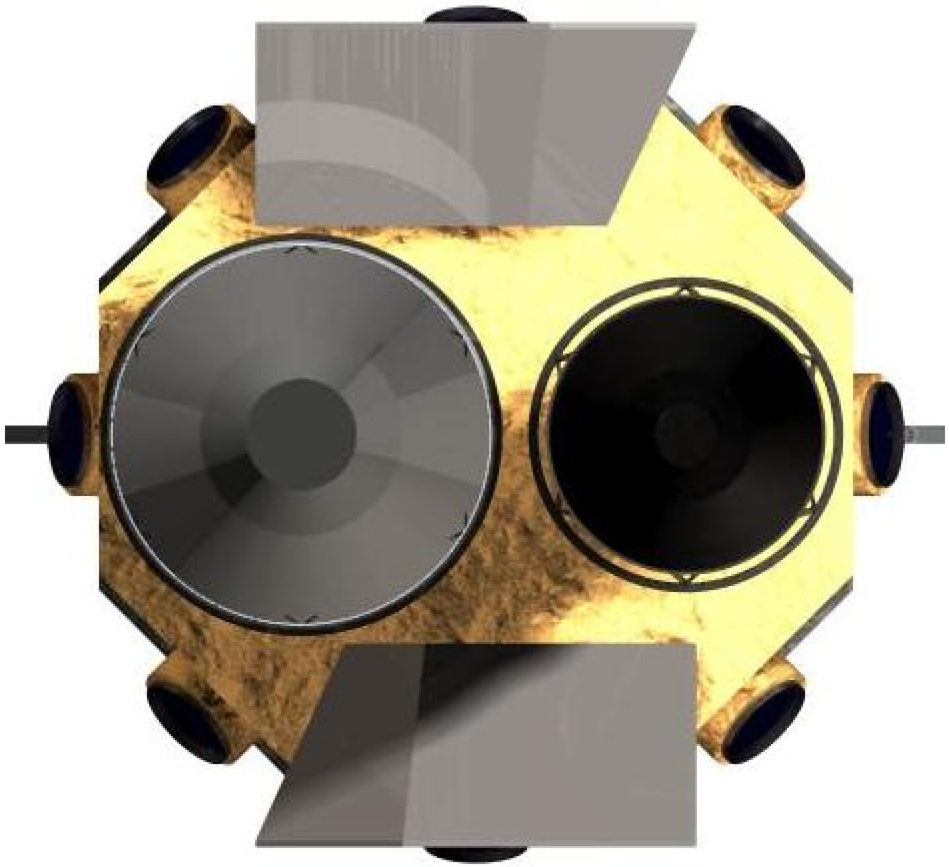
- **Array Regulation – Direct Energy Transfer (0.95 Efficiency)**

Configuration: 8 GRB Detectors



Configuration: 8 GRB Detectors

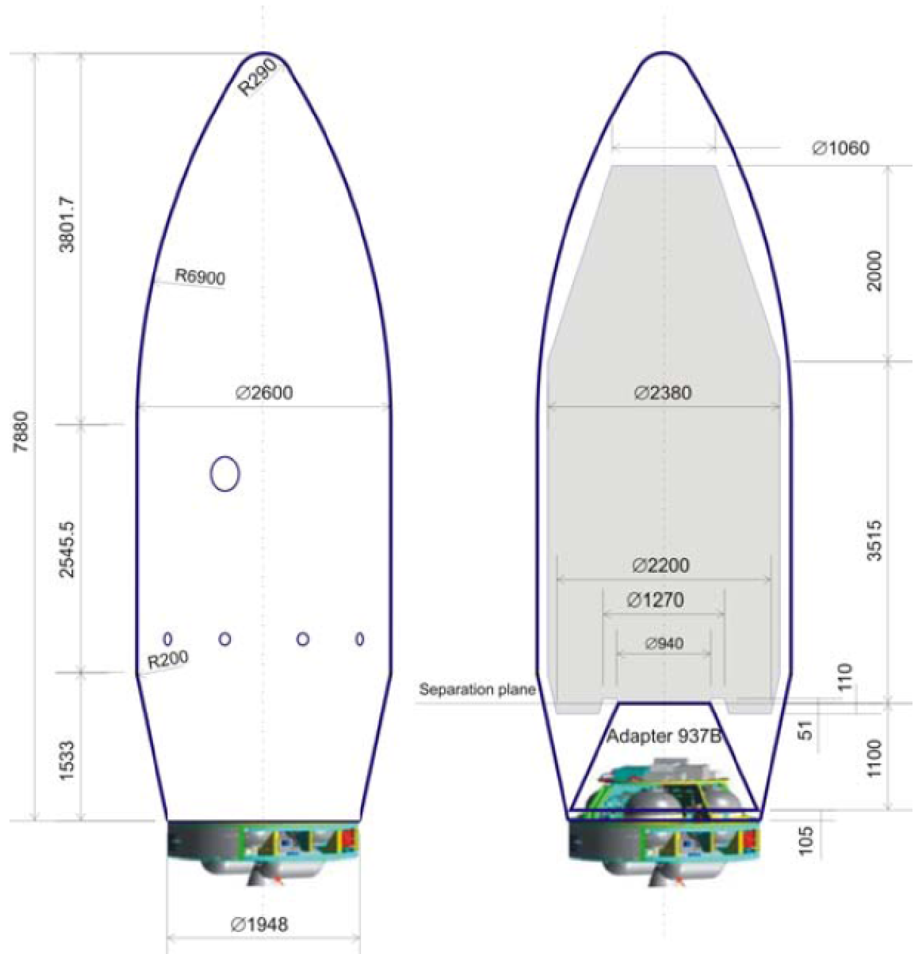
Top View



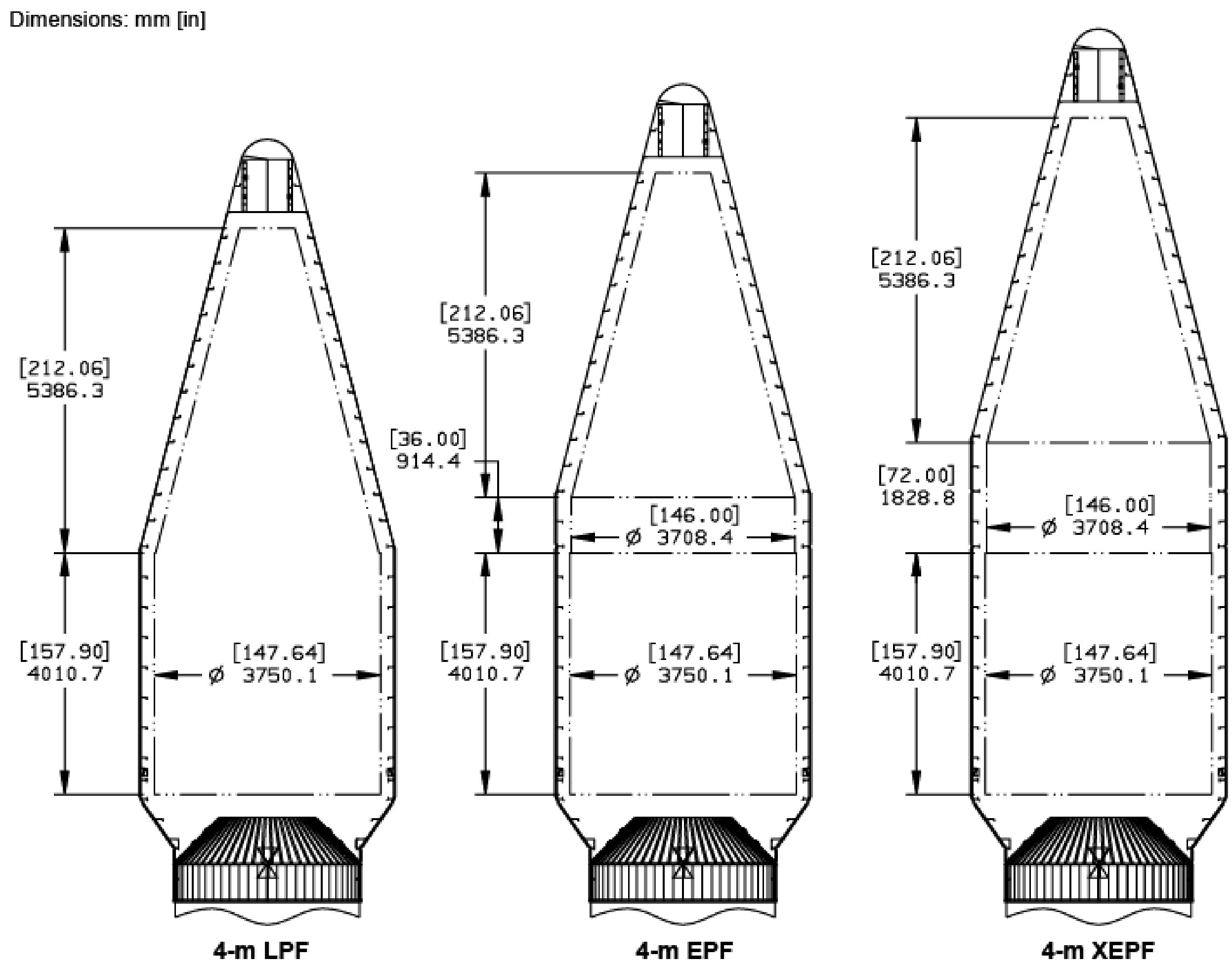
Configuration: 8 GRB Detectors

Top View, Atlas V 401 Shroud

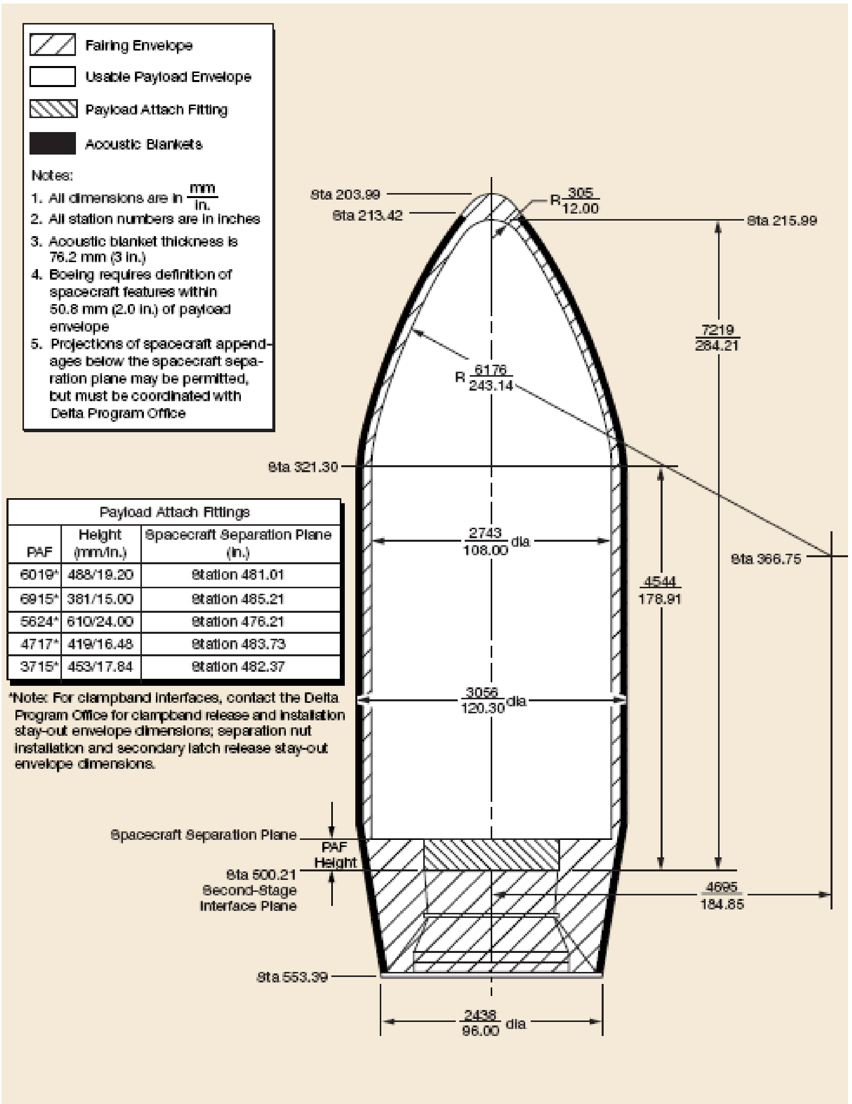
Vega Payload Fairing Dimensions



Atlas V 4 Meter Payload Fairing Dimensions



Delta-II Heavy Payload Fairing Dimensions (10ft Dia)



Payload Attach Fittings		
PAF	Height (mm/in.)	Spacecraft Separation Plane (in.)
6019*	488/19.20	Station 481.01
6915*	381/15.00	Station 485.21
5624*	610/24.00	Station 476.21
4717*	419/16.48	Station 483.73
3715*	453/17.84	Station 482.37

*Note: For clampband interfaces, contact the Delta Program Office for clampband release and installation stay-out envelope dimensions; separation nut installation and secondary latch release stay-out envelope dimensions.

Figure 3-9. Payload Static Envelope, 3-m (10-ft)-dia Fairing, Two-Stage Configuration (Various PAFs)

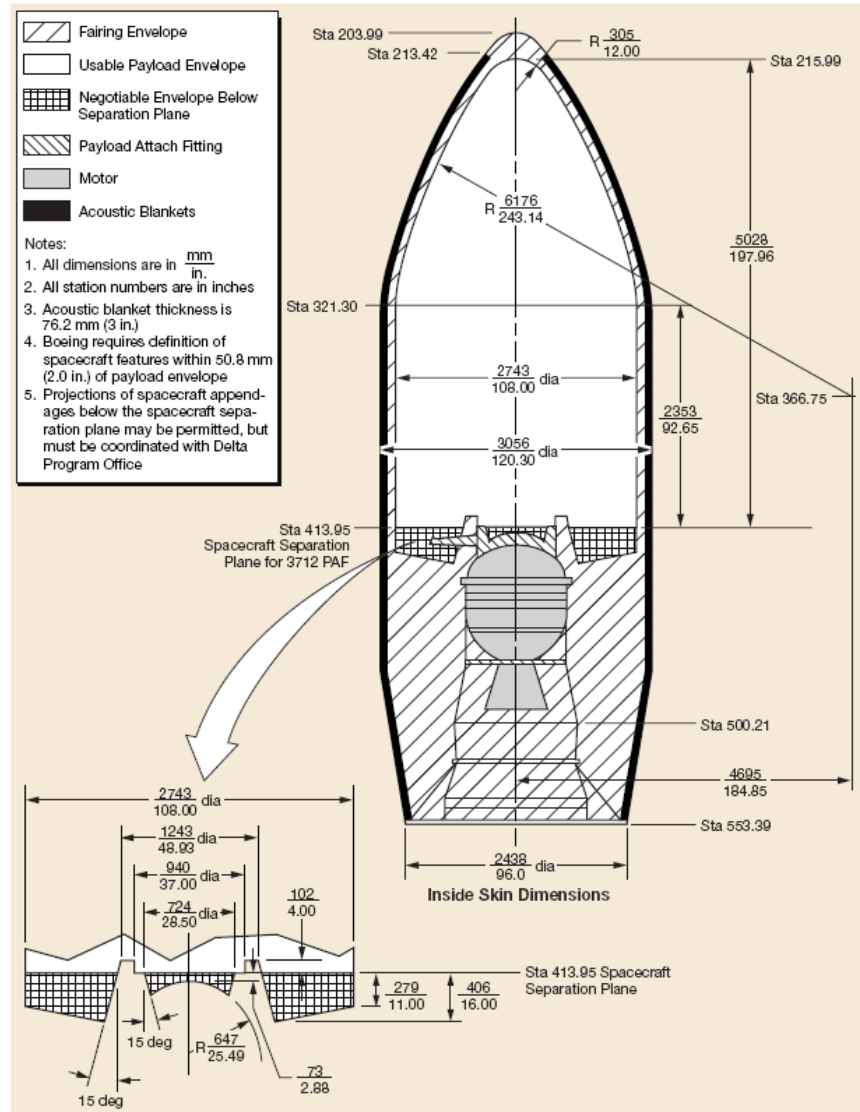


Figure 3-8. Payload Static Envelope, 3-m (10-ft)-dia Fairing, Three-Stage Configuration (3712 PAF)